III. *Viking* Project, Orbiter System and Project Support

A. Project Description and Status

1. Description

The primary objective of the *Viking* Project is to send two vehicles to the planet Mars to perform scientific experiments directed toward enhancing current knowledge about the physical characteristics of the planet, particularly its capability for supporting life and possible evidence of life. The two vehicles, each consisting of an orbiter system and a lander system, are anticipated for launch during 1975. The orbiter system will be developed by JPL, and the lander system will be developed by the Martin-Marietta Corp. Langley Research Center has overall management responsibility for the *Viking* Project.

The orbiter system will transport and inject the lander system at the appropriate point for a selected landing site and will relay telemetered data from the lander to earth. Scientific instruments on the orbiter will be used to measure atmospheric and surface parameters at various times and locations to determine the dynamic characteristics of the planet. The topography of Mars will be mapped during orbital operations, with special emphasis on mapping the proposed landing site prior to deorbit of the lander system and on supporting the lander after it is on the surface of Mars by determining if changes are apparent in the vicinity of the lander. Both visual and infrared coverage will be possible during the presently planned 140 days of orbital operations.

During entry and after landing, the scientific instruments on the lander will measure Mars' atmospheric composition, temperature, pressure, and density. After landing, the topography of the landing site will be mapped, and measurements will be made of the planet's surface composition, temperature, pressure, humidity, and wind speed. Of particular interest in the surface measurements are the type of organic compounds present, if any, and the amount and form of water. A gas chromatograph/mass spectrometer, for measuring both atmospheric and surface composition, is being developed by JPL. The surface soil analyses will be directed at detecting evidence of growth and/or metabolism.

2. Status

During this period, authorization was received from the Langley Research Center to proceed with the plans on the *Viking* 1975 mission. Accordingly, intensive efforts were initiated to prepare cost estimates for the redirected orbiter activities.

Commonality studies are being conducted with the *Mariner* Venus–Mercury 1973 Project Office to determine if such an approach is a cost-effective measure. Specific subsystem areas being investigated include flight data, TV, data storage, S-band radio, and central computer.

Significant progress was made in identifying JPL's responsibilities and overall schedule activities in mission analysis and design. A study was conducted on *Viking* 1975 navigational characteristics to provide a consistent basis for forthcoming hardware system proposals. Effort was also initiated to develop the capability to analyze the navigational accuracy requirements of broken plane trajectories.

B. Space Sciences

1. Development of an Optimal Optical Design for the Mars Atmospheric Water Detection Spectrometer

a. Introduction. A high-resolution spectrometer has been selected for the orbiter science experiment of the *Viking* 1975 mission to determine the amount and distribution of water vapor in the Martian atmosphere. The experiment proposal by Dr. C. B. Farmer, Principal Investigator, outlined the considerations leading to the choice of an instrument operating in the 1.4-μ region with a spectral resolution of no more than 0.5 cm^{-1}.

b. Design constraints. In order to measure the radiant flux from Mars at this spectral resolution, a monochromator with a relative aperture of *f*/5 and a focal length of 1000 mm was needed. The preliminary design proposed a double-cassegrain monochromator permitting the overall package length to be approximately one-half the required focal length. This made it feasible to achieve the size limit that would be reasonable for a spacecraft instrument.

Plans were to measure the intensity in four narrow frequency bands between 7231.18 cm^{-1} and 7234.06 cm^{-1}, the bandwidth to be controlled by an entrance slit 0.25 mm wide by 1.0 mm high and measured by lead sulphide detectors placed at the spectral image plane. A 300-line/mm (fourth-order), or a 1200-line/mm (first-order) grating would be needed to disperse the energy with the appropriate spectral resolution.

c. Preliminary ray-trace design. Based upon the constraints needed for the high-resolution spectrometer, a contract was awarded to Beckman Instruments, Inc., to ray-trace the cassegrain monochromator configuration as well as any other configuration capable of providing all requirements.

As reported in Ref. 1, eight other monochromator configurations were subsequently ray-traced and compared in performance and structural desirability with the cassegrain configuration. These eight systems were all double-pass monochromators; the second pass through the monochromator occurs after the first spectral image is returned by a corner reflector located in the image plane.

The eight configurations that were studied are as follows:

(1) An Ebert system, with a spherical collimating mirror and a corner return mirror.

(2) An Ebert system, with a spherical collimating mirror and a corner return prism.

(3) An Ebert system, with a parabolic collimating mirror and a corner return mirror.

(4) An Ebert system, with a parabolic collimating mirror and a corner return prism.

(5) A Littrow system, with a spherical collimating mirror and a corner return mirror.

(6) A Littrow system, with a spherical collimating mirror and a corner return prism.

(7) A Littrow system, with a parabolic collimating mirror and a corner return mirror.

(8) A Littrow system, with a parabolic collimating mirror and a corner return prism.

From the ray-trace analysis of these eight configurations, only one was of sufficiently high spectral image quality to meet the requirements. This was the Littrow double-pass with parabolic collimating mirror and corner return prism.

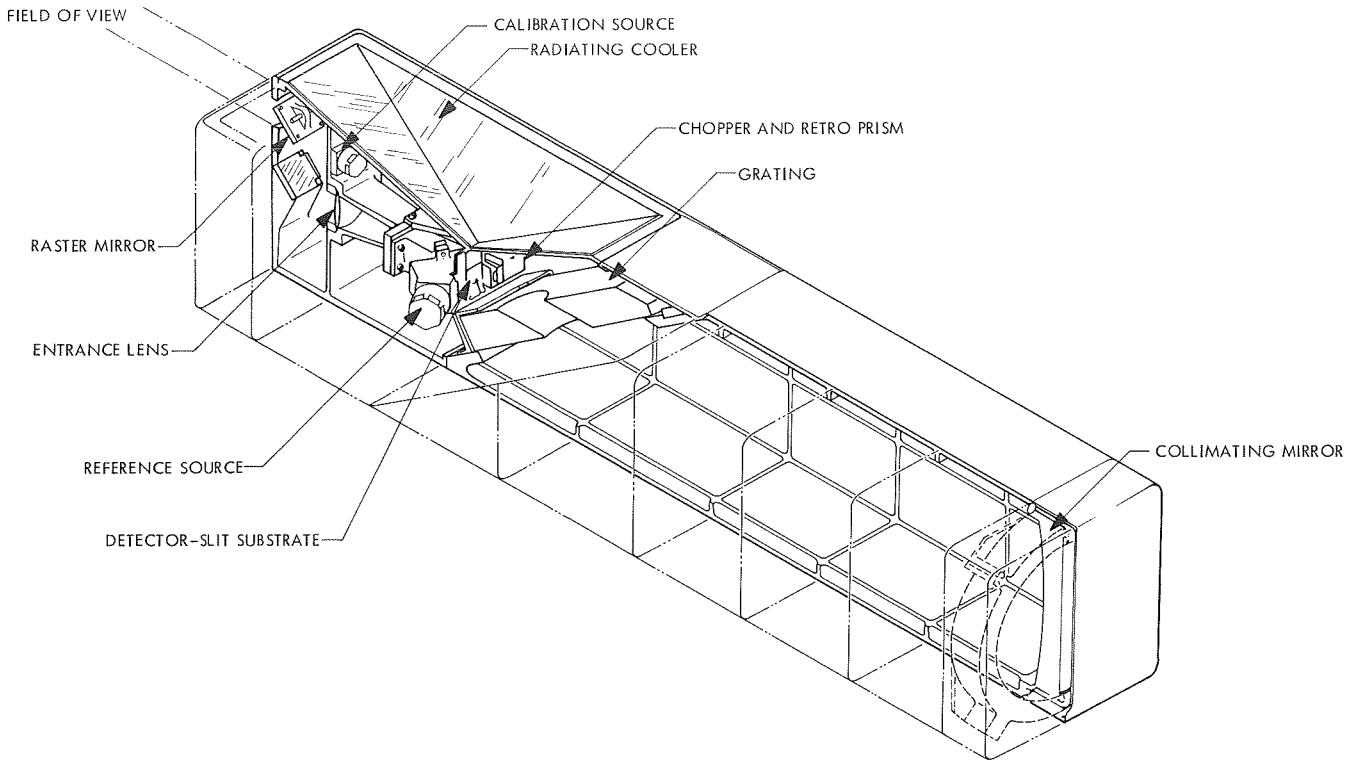

Fig. 1. Conceptual configuration of the Mars atmospheric water detection spectrometer

This configuration was selected for further analysis for the final spectrometer design. The cassegrain configuration, although a little shorter than the double-pass monochromator, would be twice as large in linear dimensions of both the grating and collimating mirror, producing a unit which would be eight times more massive. This then made the Littrow configuration even more attractive for spacecraft use.

d. Final ray-trace design. During the second phase of the contract, a ray-trace analysis was conducted on the double-pass Littrow monochromator. A final report (Ref. 2) presents a critical analysis of the optical system giving the locations of all the optical elements and optimizes both mechanical design considerations and spectral image quality.

For the final design, five new frequencies were specified instead of the four used in the preliminary study (still in the 1.4-μ region). In addition, an investigation of available line emission sources to be used for frequency calibration was undertaken. The investigation showed that a krypton arc produced a strong isolated line about 1.6 mm from the entrance slit on the side opposite from the location of the five specified frequencies.

e. Conclusions. The final report on the ray-trace study provides the necessary data for an optimal optical instrument within the capability of present-day techniques for fabrication. Based upon this ray-trace, a feasibility model was built which incorporated all the critical optical features of a flight instrument. Figure 1 shows the general arrangement of the spectrometer; Fig. 2 is a photograph of the unit.

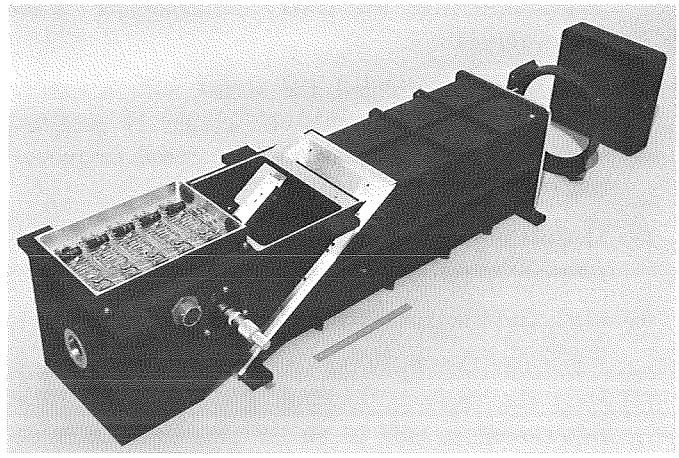

Fig. 2. Mars atmospheric water detection spectrometer

This feasibility model has demonstrated that the design concept is correct and the performance is as predicted and meets the experiment requirements. In addition, the model has pointed the way toward proper fabrication, assembly, alignment, test, and calibration techniques to be used on future instruments.

References

1. Henderson, B. D., Preliminary Report IR-2579-101. Beckman Instruments, Inc., Fullerton, Calif., May 23, 1969.
2. Henderson, B. D., Final Report FR-2579-101. Beckman Instruments, Inc., Fullerton, Calif. (undated).

C. Telecommunications

1. *Viking* Orbiter 1975 Radio Frequency Subsystem

a. Introduction. The *Viking* Orbiter 1975 radio frequency subsystem (RFS) will be basically the same as the *Mariner* Mars 1971 RFS with some relatively minor design changes. A proposed redundant receiver represents the only major change. As with previous *Mariner* radio subsystems, the *Viking* Orbiter 1975 RFS will provide the communications and tracking link between the flight orbiter and the ground-based Deep Space Instrumentation Facility (DSIF). The RFS will provide the following functions:

(1) Receive the S-band radio frequency (RF) signal transmitted from the DSIF.

(2) From the RF signal, demodulate the composite command signal and send it to the orbiter and lander command subsystems for direct commands to the orbiter and lander prior to separation.

(3) From the RF signal, demodulate the ranging signal transmitted from the DSIF and condition it for retransmission.

(4) Transmit a modulated RF signal with a carrier which is phase coherent with either the received carrier or with an internally generated frequency source.

(5) Modulate the transmitted carrier with the composite engineering and science telemetry signals.

(6) Modulate the transmitted carrier with the receiver-detected ranging signal.

(7) Send the received ranging signal to the X-band experiment as well as an RF signal which is coherent with either the received carrier or with an internally generated frequency source.

To accomplish these functions the RFS contains the subassemblies shown in the block diagram of Fig. 3. The S-band antenna subsystem [high-gain antenna (HGA), medium-gain antenna (MGA), and low-gain antenna (LGA)] is not part of the RFS and is shown for information only. The nominal parameters of the RFS are given in Table 1.

b. Receiver and diplexer subassemblies. The receiver and diplexer subassemblies, when normally interconnected, form the narrow band, double conversion, automatic phase and frequency tracking spacecraft RFS receiver. When the receiver is phase-locked to the transmitted uplink signal, it supplies an output frequency to the exciter which is phase and frequency coherent to the received signal. The exciter multiplies the frequency and phase by 120 for transmission at S-band. In this mode, the receiver and exciter form the transponder which coherently translates the frequency and phase of the received signal by a fixed ratio of 240/221. The transponder provides for coherent two-way doppler tracking of the spacecraft which permits an accurate method of determining the spacecraft trajectory. When the receiver is phase-locked to a received signal, which has been

Table 1. Nominal values of significant RFS parameters

Parameter	Value
Receiver input frequency, MHz	2115±5
Receiver tracking threshold, dBmW	−153
Receiver tracking loop noise bandwidth at threshold, Hz	18
Command channel noise bandwidth (single-sided), kHz	2.5
Ranging channel noise bandwidth (single-sided), MHz	1.5
Receiver noise figure, dB	7.5
Dynamic signal level range, dBmW	−70 to threshold
Transmit/receive frequency ratio	240/221
Transmitter output frequency, MHz	2295±5
Transmitter output power, W	
Low-power mode	10
High-power mode	20
Telemetry and ranging channel modulation bandwidth, MHz	1.5
Power required, W	
2.4 kHz 50 V	30
25–50 Vdc TWT HP/LP	92/56
Weight of radio assembly (including case and case harness assembly), lb	63

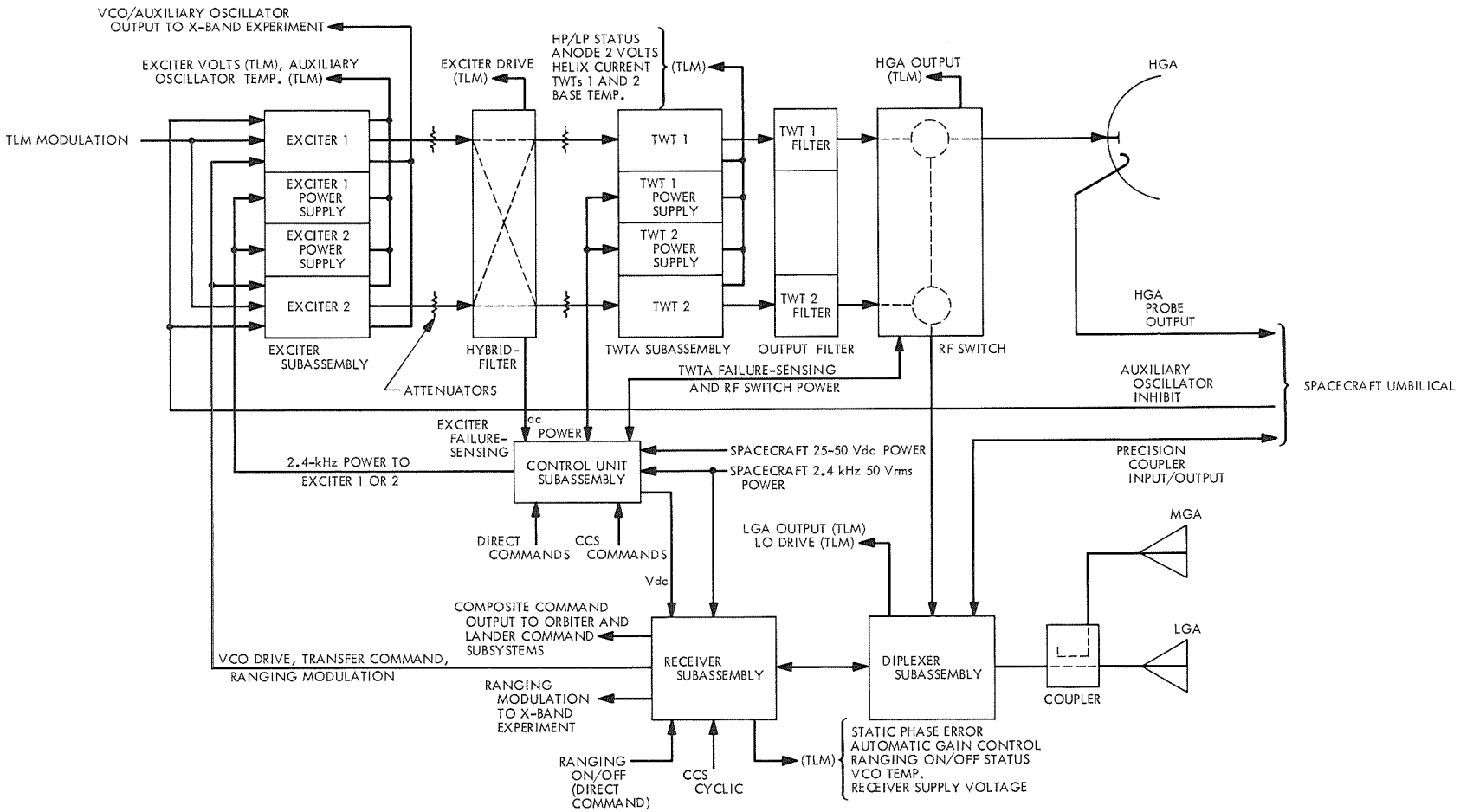

Fig. 3. *Viking* orbiter radio frequency subsystem

phase-modulated with the composite command signal, the receiver demodulates the signal and sends it to the orbiter command subsystem for direct commanding of the orbiter spacecraft. For *Viking* 1975 a second output is provided for the lander command subsystem. This allows commanding of the lander prior to separation via the orbiter S-band link. The receiver also demodulates the ranging code from the uplink signal, amplifies and limits it, and sends it to the exciter for modulation on the downlink carrier. This permits precise measurement of the distance and rate of change of distance to the spacecraft. For *Viking* Orbiter 1975 a separate ranging output will be provided for the X-band experiment (see *Paragraph g*).

The receiver has six analog telemetry (TLM) signals sent to the flight data subsystem. AGC 1 (TLM) and AGC 2 (TLM) indicate the level of the spacecraft received carrier; static phase error (TLM) indicates the operating frequency of the receiver voltage-controlled oscillator (VCO); LO drive (TLM) indicates the drive level of the first local oscillator (LO) into the first mixer. The receiver supply voltage (TLM) is new for *Viking* 1975 and will indicate the relative level of the ±15-Vdc receiver supply. A temperature transducer located on the VCO module provides a telemetry indication of the VCO temperature (TLM) and allows prediction of the temperature variable VCO best lock frequency. (Best lock frequency is the ground transmitter frequency which has the highest probability of obtaining receiver phase lock in the shortest amount of time.)

The receiver has a self-contained power supply which converts the orbiter 2.4-kHz power to the dc voltages required by the receiver. A circuit breaker in the supply protects the 2.4-kHz spacecraft power when an excessive current load is drawn by the receiver. If the receiver draws a current greater than a preset level, the receiver power is turned off. Since the overcurrent could be momentary, a computer command subsystem (CCS) cyclic will reset the circuit breaker only if the excess current load is no longer present. The supply also provides a voltage to the first stages of the receiver video amplifier module. This voltage turns the ranging channel *on* or *off* by a direct command or *off* by a CCS cyclic. The channel status is indicated by a voltage sent to a telemetry status channel to indicate whether the ranging channel is *on* or *off*. The turning *off* of the ranging channel by the CCS cyclic is a safeguard against the feedback observed on the *Mariner* Mars 1964 RFS which created "self-lock" and was observed briefly and noncatastrophically in the *Mariner VI* receiver. "Self-lock" can cause performance degradation or a complete loss of command and two-way tracking capabilities.

c. Exciter subassembly. The exciter subassembly consists of two identical and fully redundant S-band exciters. The output power of each exciter is divided in half by the hybrid-filter and presented to the input of both of the redundant traveling-wave tube amplifiers (TWTAs). Thus, the exciters and TWTAs provide a fully redundant transmitter chain. The exciters are phase-modulated by the composite telemetry signal for transmission of scientific and engineering information from the orbiter. If the receiver ranging channel is *on*, the exciter carrier is also modulated with the ranging signal from the receiver. The exciter output signal is derived from either an internal crystal-controlled oscillator or from the receiver VCO. When the receiver is phase-locked to an uplink carrier, a switching voltage from the receiver inhibits the internal oscillator and allows the VCO to drive the multiplier chain for coherent two-way doppler tracking. When the receiver is out of lock, the receiver switching voltage changes and enables the internal auxiliary oscillator. The *Mariner* Mars 1971 RFS has added a spacecraft umbilical control which inhibits the transfer from VCO to the internal auxiliary oscillator. This is used to measure the free-running (unlocked) VCO frequency on the downlink, which allows the measurement of best lock frequency in test and on the launch pad. For *Viking* Orbiter 1975 a separate output from each of the two exciter auxiliary oscillators is provided for the X-band experiment (see *Paragraph g*).

The exciter power supply converts the spacecraft 2.4-kHz power to the dc power required by the three exciter modules. Two fully redundant supplies are housed in one module; one supply is *off* and one supply provides power to only one exciter. Switching of exciters is accomplished in the control unit, which switches the 2.4-kHz power to either of the supplies. The switching is accomplished in one of three ways: direct command or two failure modes. If the exciter draws excessive 2.4-kHz current, the overcurrent sensor in the control unit automatically switches to the other exciter. Also, a diode in the hybrid-filter samples the exciter S-band output signal and provides an output voltage to the control unit proportional to the RF power level. If the power level drops below a preset level, a CCS cyclic will initiate a switch to the redundant exciter. A second power detector provides the voltages proportional to the exciter output for exciter drive telemetry.

The telemetry relay module is common to both exciters. It provides the functions of conditioning the composite telemetry signal before reaching the phase modulator and of switching between two telemetry functions depending on which exciter is *on*. One of the switched telemetry signals is a monitor of the dc power supply voltage. This voltage is conditioned in each power supply so that the returned telemetry will indicate which exciter is operating. The temperature transducers located on each of the auxiliary oscillator modules are also switched. This temperature measurement allows accurate prediction of the temperature variable downlink auxiliary oscillator frequency.

d. Control unit. The control unit accepts direct commands from the orbiter command subsystem, the computer command subsystem (CCS), and the failure-sensing circuits in the RFS. From these inputs it selects the appropriate RFS operating mode.

The functions of the control unit are:

(1) To switch the transmitter output to the high-gain or low-gain antenna upon receipt of direct or CCS commands.

(2) To sense the TWTA and exciter power output level and, if either is low, to switch to the redundant unit upon receipt of the CCS cyclic command.

(3) To sense overcurrent conditions in the TWTA or exciter power supplies and switch immediately to the redundant unit, and to switch to either element upon receipt of a direct command.

(4) To select high- (HP) or low-power (LP) output from the TWTA upon receipt of direct or CCS commands.

(5) To supply switching logic voltage to the receiver power supply ranging and overcurrent functions.

e. Traveling-wave tube amplifiers. The traveling-wave tube amplifier (TWTA) consists of two Hughes Aircraft Co. Model 242H traveling-wave tubes (similar to the Model 394H used on the *Apollo* program), each with its encapsulated, modular, solid-state power supply. Both TWTAs are mounted in a single case.

The TWTA functions as a power amplifier of the phase-modulated S-band signal from the exciter. It provides two output power levels, nominally 10 or 20 W, by changing the voltages applied to the TWT helix, collector, and first anode. A timer delays application of high voltage to the TWT for 90 to 150 s to allow the heater to warm up.

In each of the two independent dc-to-dc power supplies, the primary 25- to 50-Vdc line voltage is converted to a stable 20 V, which is then converted to voltages suitable for the TWT filament, first and second anodes, helix, and collector. Only one supply is *on* at a time and supplies power to only one TWT. The stable 20 V is also used to control the circulator switches in the transmitter output circuits via the control unit and to provide the voltage for failure-sensing of TWTA output power in the control unit.

A feature added to the *Mariner* Mars 1971 spacecraft will be retained for *Viking* Orbiter 1975. By direct or CCS command, the raw 25- to 50-Vdc power can be turned *on* or *off* to the RFS via the power subsystem. When the power to the TWTs is turned *off*, to conserve spacecraft power or to prevent RF interference with future S-band missions, some of the power is routed to heater resistors to maintain a thermal balance in the RFS and the spacecraft.

The TWTA functions which are telemetered are helix current, second anode voltage, and a high/low-power mode status indicator. The second anode voltage is conditioned to indicate which of the redundant units is operating. The TWTA output level, which is sampled by a diode detector in the RF switch, provides a voltage proportional to the RF level to the control unit. If the power drops below a preset limit, a CCS cyclic will initiate a switch to the redundant TWTA. The TWTA has a provision for failure-switching in the event of an overcurrent as well as a backup set of fuses on the raw dc power line. If a TWTA draws current above a preset level, as sensed in the control unit, the overcurrent sensor automatically switches to the other TWTA. The TWTs can also be switched by a direct command.

f. Microwave components. The microwave components consist of the hybrid-filter, output filter, RF switch, and diplexer. The electrical characteristics are shown in Table 2. The hybrid-filter, RF switch, and diplexer use stripline on polyphenylene oxide construction for the hybrid and all power monitors. Coaxial cavities are used in the other circuits.

Table 2. Electrical characteristics of microwave components

Component	Insertion loss, dB		Rejection level, dB	Reverse isolation, dB	
	Transmit[a]	Receive[b]		Minimum	Ambient (25 ± 5°C)
Hybrid-filter	4.5	>100	>100 (dc to 2117 MHz) >60 (2199.8 and 2391.8 MHz) >100 (2477 to 4000 MHz) >70 (4000 to 10,000 MHz)		
Output filter	<0.3	>60	>30 (2100 and 2125 MHz) >70 (4580 to 10,000 MHz)		
Diplexer					
LGA to receiver	>100	<1.2	>100 (dc to 1988.4 MHz) >70 (1988.4 to 2048.4 MHz) >70 (4580 to 10,000 MHz)		
Transmitter to LGA	<0.3				
Transmitter to receiver		>50			
RF switch					
TWTA 1 to HGA	<0.35			>20	>30
TWTA 2 to HGA	<0.7			>40	>60
TWTA 1 to LGA	<0.7			>40	>60
TWTA 2 to LGA	<0.35			>20	>30

[a]2290.2 to 2301.3 MHz.

[b]2108.7 to 2119.2 MHz.

The hybrid-filter divides the power output from each exciter and furnishes drive power to both TWTs simultaneously through band-pass and low-pass filters. The hybrid-filter also contains two diode monitors for telemetry and failure mode sensing of the exciter RF output level. Attenuators are used in the hybrid-filter input and output circuits to adjust the exciter drive to an optimum value for each TWTA, and to compensate for hybrid-filter unbalance.

The output filter is used in the TWTA output circuit; it has minimum insertion loss at transmitter frequencies, a band-reject filter at receiver frequencies, and a low-pass filter to attenuate transmitter output harmonics.

The RF switch consists of two ferrite RF circulators making a four-port circulator switch. It provides a means of connecting either TWT to the high-gain or the low-gain antenna. A permanent magnet assures that TWTA 1 will be connected to the high-gain antenna with about a 3-dB power loss in case the dc switch coil voltage supply fails. Failure-sensing for the output of each TWT is accomplished at the input to each circulator by diode power monitors, and the high-gain antenna telemetry power monitor is at the high-gain antenna output of the switch.

The diplexer provides a band-reject filter at receiver frequencies between the transmitter output (RF switch) and the receiver input and a band-pass filter (at the receiver frequency) and a low-pass filter between the low-gain antenna and receiver (first mixer) input. The diplexer also has a precision (directional) coupler. This coupler provides the means for measuring transmitter output power to the low-gain antenna. It also provides a calibrated path to the receiver for uplink receiver measurements, and orbiter commanding during ground testing. Access to the precision coupler input/output is provided via the spacecraft umbilical connector.

g. S-X-band experiment interface. One of the changes to the *Mariner* Mars 1971 RFS design for *Viking* Orbiter 1975 is to provide a ranging and VCO/auxiliary oscillator output to the X-band experiment. The X-band experiment will transmit via the S-X-band high-gain antenna a nominal 8415-MHz signal (compared to the 2295-MHz S-band signal). By comparing the doppler and ranging measurements of the X- and S-band signals, information

about the transmission media between the orbiter antenna and the ground antenna will be obtained (e.g., space electron charge densities, Mars and earth ionospheres, earth and Mars neutral atmospheres, and solar atmosphere).

Figure 4 shows a block diagram of the interface between the RFS and the X-band experiment. The system as shown will allow S- and X-band tracking in three modes and will allow either of the redundant auxiliary oscillators to drive the X-band transmitter. The two primary modes are two-way tracking and two-way tracking with ranging. When the RFS is in two-way track, the receiver VCO drives the S- and X-band transmitters. Doppler deviations between the S- and X-band signals can be coherently compared to the signal transmitted to the spacecraft from the DSIF. When the ranging code is being modulated on the uplink during two-way S-band tracking, the receiver demodulates, amplifies, and limits the ranging signal. This ranging signal is then modulated on the S- and X-band carriers. Range deviations between the two signals can then be compared. The third mode is when the RFS is in one-way track. For this case the internal auxiliary oscillator of the RFS drives the S- and X-band transmitters. This provides two signals coherent to the same source for measurement of one-way doppler deviation of the two signals.

The isolation switch, shown between the RFS auxiliary oscillator outputs and the X-band multipliers, has been added to prevent possible VCO leakage from the *off* auxiliary oscillator from modulating the X-band multiplier. Special tests on the VCO isolation of an unpowered auxiliary oscillator may show that this switch is not required.

As presently planned, the X-band experiment is as shown in Fig. 4. The RFS VCO or auxiliary oscillator output frequency is multiplied by four and then phase-modulated by the ranging signal (if present) from the RFS. The times 10 and times 11 multipliers provide an additional multiplication of 110 for a total multiplication of 440. The output of the final multiplier is expected to be about 200 mW at 8415 MHz. The X-band signal only drives the dual-feed S-X-band high-gain antenna.

h. Redundant RFS receivers. Because of mission dependence on command and two-way tracking, as well as increased operating lifetime from *Viking* 1973 to *Viking* 1975, a study of redundant RFS receivers was initiated.

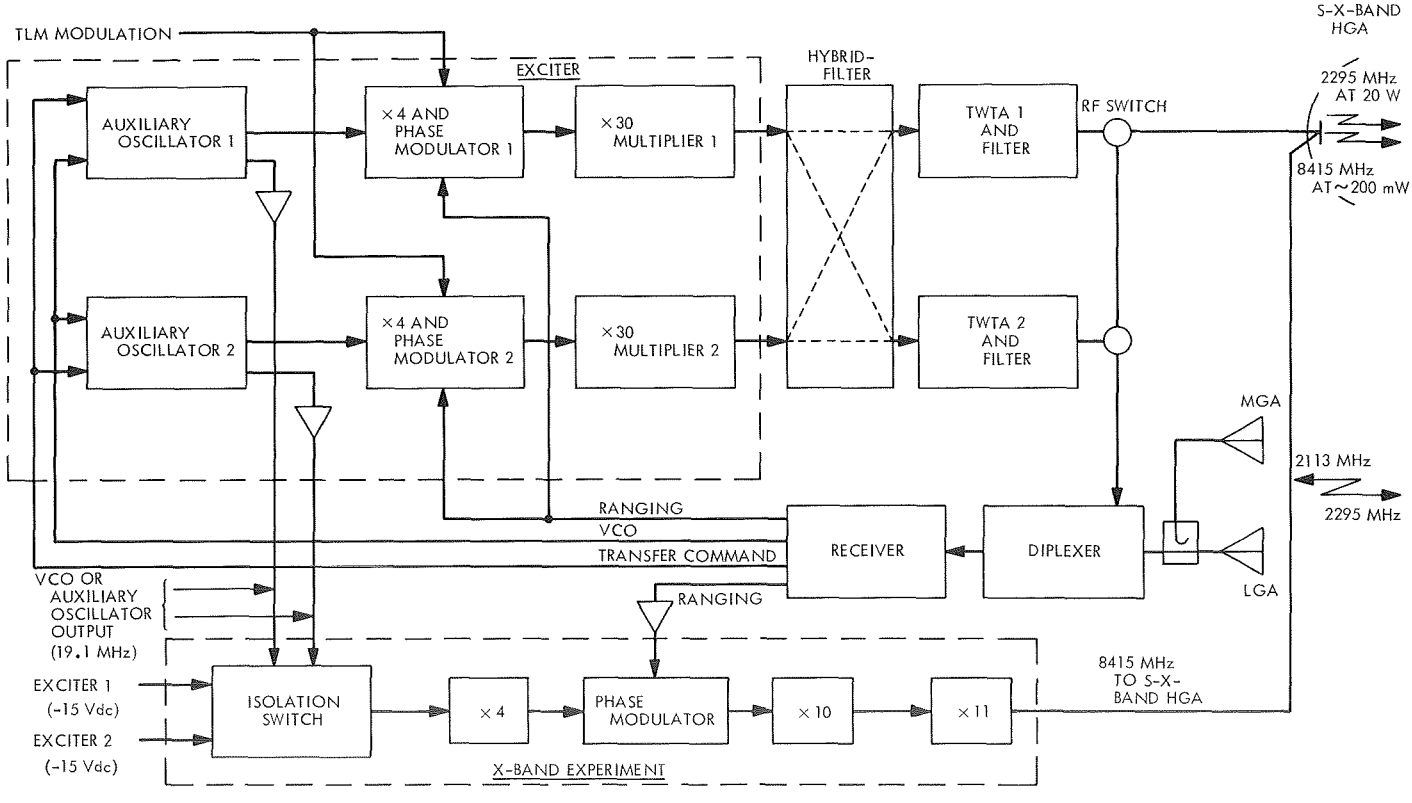

Fig. 4. S-band RFS and X-band experiment interface

This would represent a major design change to the *Viking* Orbiter 1975 RFS. The study on redundant receivers has also instigated a study of the overall redundancy in the telecommunications subsystems. This includes redundant receivers, command detectors, telemetry modulators, and the existing redundant transmitters. Since it is difficult to discuss RFS redundancy without involving command or telemetry, Figs. 5 and 6 include the telemetry modulators and the command detectors. Figure 5 represents the simplest form of redundancy, which might be called "functional" redundancy. In this case one receiver, command detector, telemetry modulator, exciter, and power amplifier make up one "functional" receive/transmit chain. The redundant chain is identical and separate. There is no cross coupling among the redundant elements. If any one element fails, the entire functional backup chain must be selected for full receive/transmit capability. Figure 6 represents a more complex form of redundancy, which could be called "block" redundancy. In this case each "block" or subassembly, such as an exciter or a command detector, can be individually selected to operate. This form of redundancy provides the maximum cross coupling between redundant elements. If any one element fails, just its redundant element need be switched. For example, the present redundant transmitter chain shown in Fig. 3 represents a form of "block" redundancy; either TWTA or either exciter can be selected if one element fails.

A number of tradeoff and reliability studies remains to be considered prior to selection of the final design.

2. *Viking* Orbiter Relay System

a. Introduction. The *Viking* Orbiter Relay System provides receiving and detection capability for communications via a UHF lander–orbiter link. This system operates at various times in the mission with the orbiter acting as a data relay point between the lander and earth. The lander data is pulse-code-modulated–frequency-shift-keyed (PCM–FSK) at one of two available bit rates (4 or 16 kbits/s). The received data is recorded in the orbiter and, in the case of the 4-kbits/s transmission, may also be relayed to earth in real time via the orbiter–earth S-band link. Recorded data is relayed to earth via S-band at a later time.

The Orbiter Relay System encompasses two electronic subsystems and a UHF antenna. These are:

(1) Relay radio subsystem.

(2) Relay telemetry subsystem.

(3) Relay antenna subsystem.

Operation of the Orbiter Relay System is initiated by ground command and/or computer command subsystem (CCS) command.

b. Relay link requirements. The Orbiter Relay System is required to support the relay link during the following mission phases:

(1) Lander–orbiter separation to touchdown.

(2) Touchdown to first setting of the orbiter.

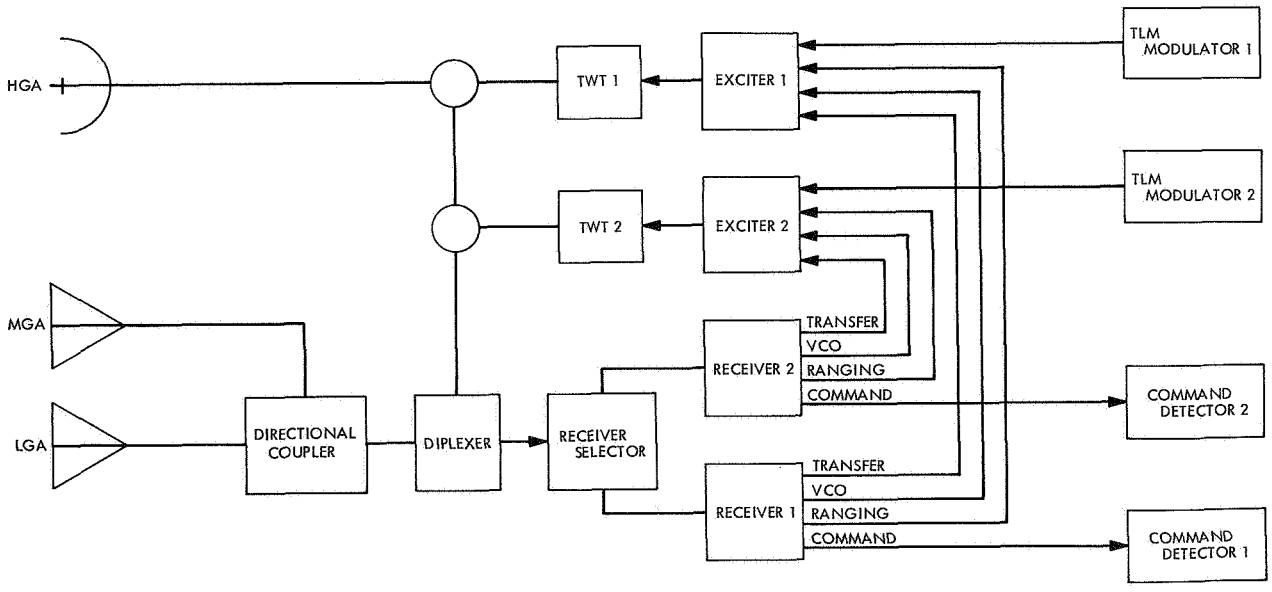

Fig. 5. *Viking* 1975 telecommunications "functional" redundant configuration (minimum cross coupling)

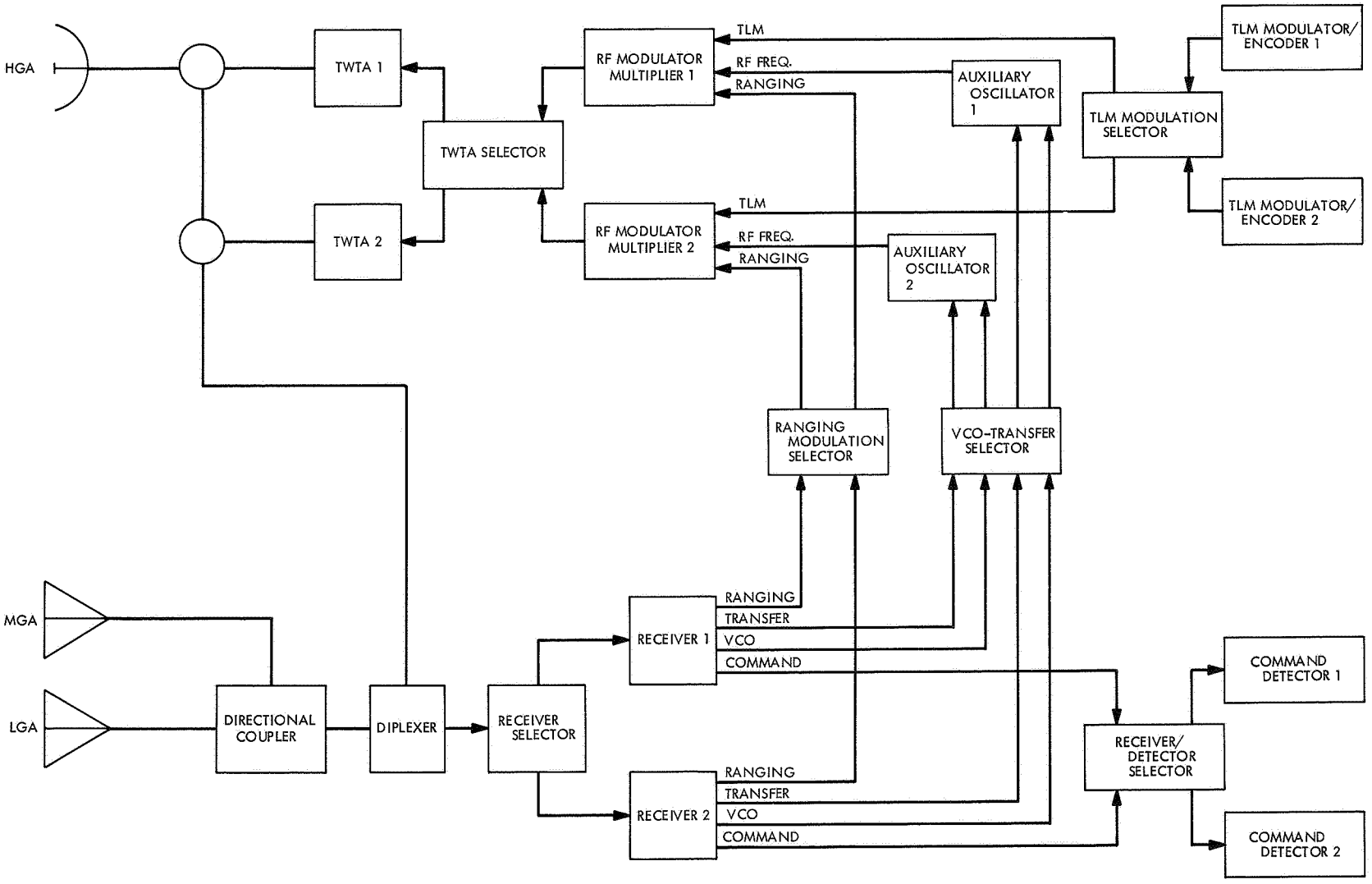

Fig. 6. *Viking* 1975 telecommunications "block" redundant configuration (maximum cross coupling)

(3) Once each pass for the first five days after landing.

(4) At least once every 30 days for the sixth through ninetieth days after landing.

If required by the lander, Phase 3 may be extended up to 90 days after landing.

During Phase 1, the lander transmits data at 4 kbits/s with 10 W of RF power. Transmission is continuous from separation to the end of de-orbit burn, then intermittent until 30 min prior to entry, and then again continuous until touchdown. Data received at the orbiter is recorded in the data storage subsystem (DSS) and also relayed in real time to earth via S-band.

Phase 2 begins at lander touchdown. The lander immediately switches to a 16-kbit/s transmission with 30 W of RF power. Data received at the orbiter is recorded in the DSS until the link drops below threshold as the orbiter sets. At least 10^7 bits of data is required in this phase.

During Phase 3, the relay link is operational once each day as the orbiter passes over the lander near periapsis. Data is sent at 16 kbits/s with 30 W of lander RF power for recording in the orbiter. This phase requires recording of at least 10^7 bits during each pass.

Phase 4 requires the relay link to be operational for at least one pass near periapsis every 30 days. These passes are a product of opportunity as the landing site occasionally comes into view to the orbiter. Normally, the link would be operated in this phase only when the orbiter's elevation angle is above 15 deg and lander–orbiter range is less than 5000 km. These passes will be supported by the lander at 4 kbits/s and 30 W of RF power.

c. Relay link performance estimates. The relay link is designed to support Phases 1 through 4 with a single configuration. Ordinarily the link would be operated with elevation angles of 15 deg or higher and ranges below 5000 km. For a periapsis pass during the first 5 days after landing, the nominal performance is estimated to be as shown in Fig. 7. This curve is based on the parameters of Table 3 for a 16-kbit/s transmission. Curves bounding the nominal for best- and worst-case multipath variations are included on Fig. 7 (a specular post-landed multipath model is assumed with relative permittivity $\epsilon_r = 3.0$). The multipath will cause rhythmic variations in margin within the envelope shown on Fig. 7. These variations are a function of elevation angle and

Table 3. *Viking* 1975 relay link parameters[a]

Parameter	Value
Transmitter power (lander), W	30
Transmitting antenna gain (peak), dB	4.5
Receiving antenna gain (peak), dB	4.5
System noise temperature, °K	618
Threshold ST_b/N_0, dB	12.3
Periapsis altitude, km	1400
PER (true anomaly of landing site), deg	−11.3
Cross-ranging, deg	0
Lander slope, deg	0

[a]High-rate data at 16 kbits/s.

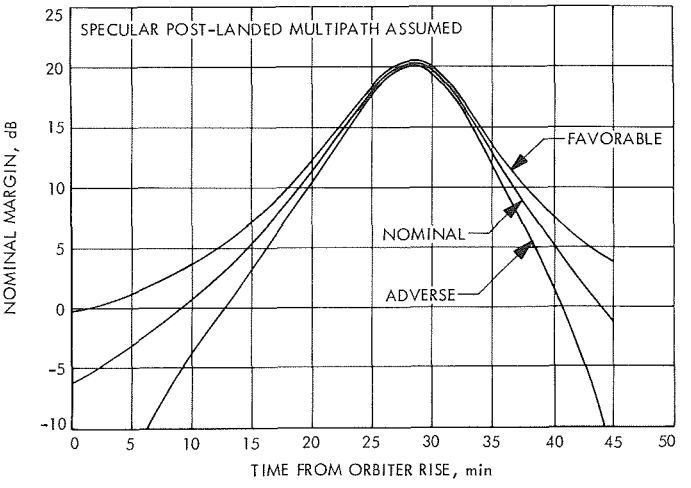

Fig. 7. *Viking* 1975 relay link performance curve showing multipath envelope

antenna height above the surface. Including actual multipath effects for an antenna height of 1 m produces the performance curve of Fig. 8.

The time above horizon and peak performance is critically dependent on the trajectory of the orbiter. Figure 9 shows performance curves for the nominal and bounding periapsis altitudes of 1000, 1400, and 1800 km. The lower the periapsis altitude, the lesser time the orbiter is in view. In order to meet the requirement of 10^7 bits per pass, the link must be above threshold for at least 10.42 min. Hence, the lower altitudes place the most severe limitation on the link from a time standpoint.

d. Beacon considerations. The Orbiter Relay System can be commanded into operation by ground or CCS command. In order to activate the lander equipment, it

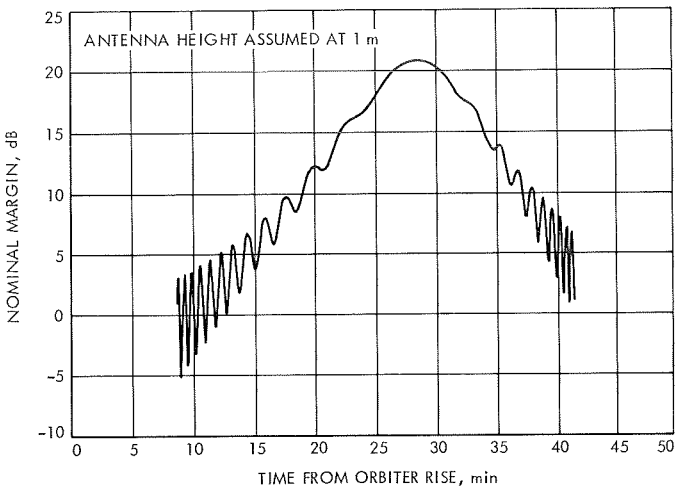

Fig. 8. *Viking* 1975 relay link performance with post-landed multipath

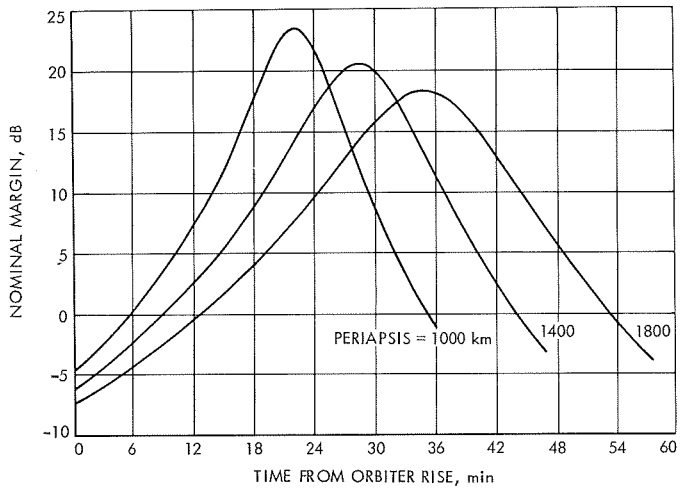

Fig. 9. *Viking* 1975 relay link performance vs time

is proposed to implement a beacon transmitter in the orbiter. This transmitter would be commanded *on* prior to orbiter rise. An address tone would be amplitude modulated on the beacon to select the desired lander. When detected at the lander, UHF transmission would begin with real-time data. After data lock is achieved in the orbiter, the address tone would be removed and an enable tone sent to the lander. The lander would then initiate playback of recorded data.

Another scheme for link control would be use of a beacon transmitter with address tones as above, but no enable tone. A fixed time delay would be programmed into lander operations to allow sufficient time for orbiter data lock. It may even be possible to devise a scheme whereby internally programmed timing would control both the lander and orbiter with no beacon at all. These possibilities are presently under study as both primary and back-up modes of operation.

e. Relay radio subsystem. The Relay Radio Subsystem (RRS) supports relay link communications by providing:

(1) The receiving, detecting, and routing of the frequency-shift-keyed lander data.

(2) The amplitude-modulated beacon signal to initiate post-landed RF transmission by the lander.

(3) Selectable beacon modulation frequencies to allow addressing any lander and to advise the lander of the lock status of the relay telemetry subsystem (RTS) bit synchronizer.

(4) A telemetered measure of received signal strength to facilitate link performance analysis.

Figure 10 depicts the major elements for deriving the RRS transmitting and receiving functions. The transmitter signal, nominally at 405 MHz, is derived by multiplying, modulating, and amplifying the output from the crystal oscillator. The oscillator must be temperature-compensated and may possibly require an oven to achieve the required frequency stability. The transmitter power amplifier must be capable of being turned *on* or *off* by the orbiter computer command subsystem (CCS). The power amplifier *on–off* control allows the transmitter to be operated in a standby mode for warm-up and frequency stabilization purposes. The leakage signal through the *off* power amplifier can also allow the standby mode to be used for pre-separation checkout of the lander relay communications system. The modulator could be capable of supplying any one of five modulating frequencies (tones). Selection of any one of four lander addressing tones could be by CCS command, while the fifth tone would be selected whenever the RRS receives the RTS signal indicating bit synchronizer "in lock."

The receiver frequency is nominally at 385 MHz. It is a single conversion type employing a low-noise preamplifier, mixer, bandpass filtering, and limiting in the intermediate frequency (IF) stages. The IF signal is routed to two demodulators, each of which employ bandpass filters (BPFs) and square-law detectors centered at the mark and space frequencies.

The demodulator filter bandwidths must accommodate: (1) possible frequency drift of the lander transmitter, (2) possible frequency drift of the RRS receiver local oscillator, (3) the maximum possible doppler frequency

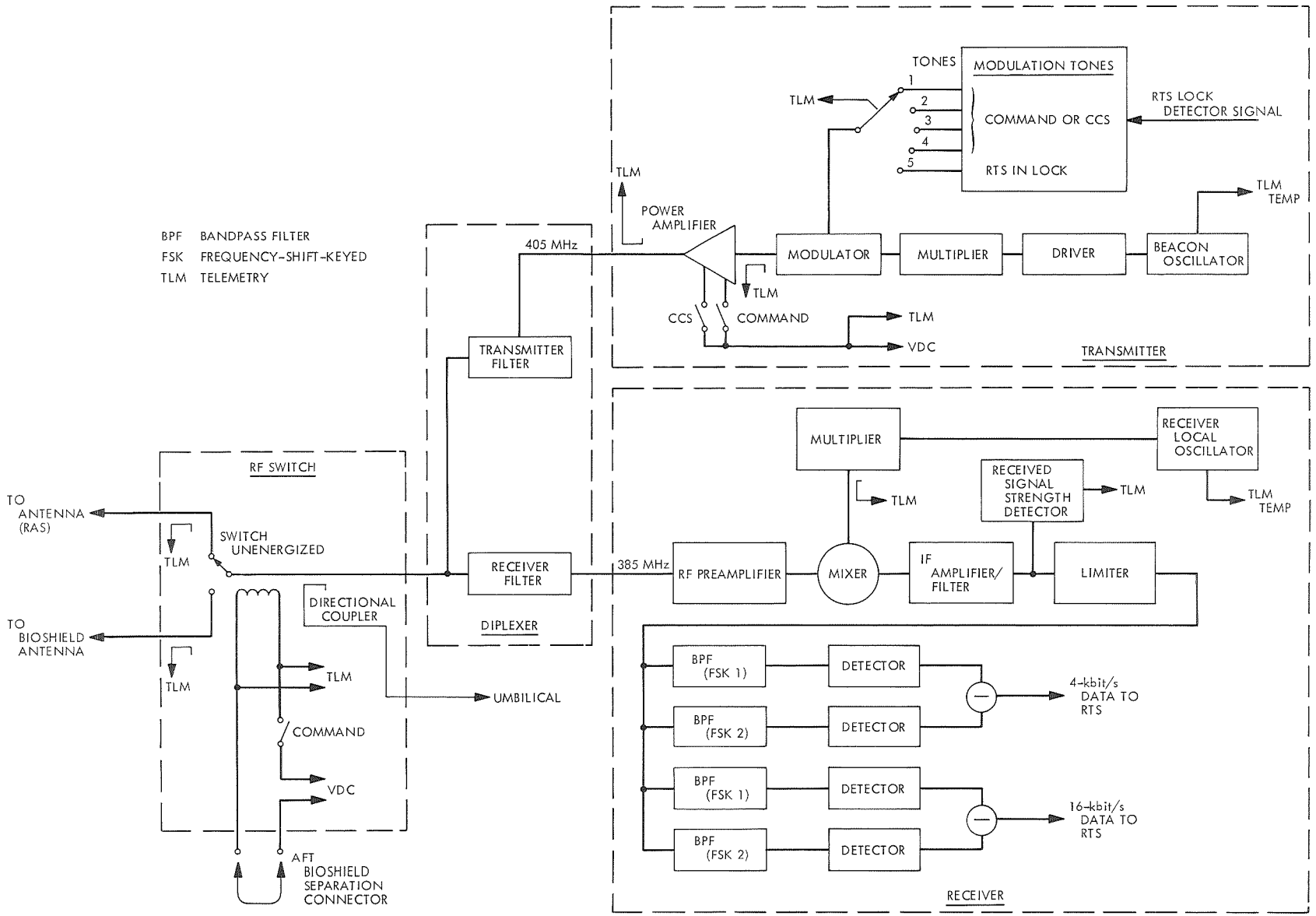

Fig. 10. Relay radio subsystem

shift between the lander and orbiter, and (4) the information bandwidth of the data. Separate demodulators for high-rate (16 kbits/s) and low-rate (4 kbits/s) data retrieval allow each demodulator to employ optimum bandwidths for that specific data rate. The mark and space filter bandwidths for the high-rate demodulator are nominally 110 kHz while the low-rate demodulator employs nominal 55-kHz bandwidths. Because the lander transmits the same mark and space center frequencies regardless of the data rate, little is gained by employing independent high-rate and low-rate IF bandwidths. Hence, the IF bandwidth, nominally 320 kHz, is determined by the same factors that determine the demodulator bandwidth plus the frequency separation between the mark and space frequencies.

The receiver design will incorporate circuitry for measuring the received signal strength at the orbiter. This measurement will be sent to the orbiter flight data subsystem (FDS) to be telemetered to earth. Current plans for implementing this measurement require positive signal-to-noise ratios (SNRs) in the receiver IF stages. At link threshold (a bit error rate of 3×10^{-3}), the SNR in the IF drops below zero dB, making the measurement limited to strong signal values. Performance analyses at weaker signal levels can be made from the RTS signal-to-noise ratio estimate.

The receiver and transmitter are connected to a common transmission network by means of a diplexer. An RF switch located at the diplexer transmit/receive terminal allows the RRS to be connected to the relay antenna or to a pickup probe mounted in the capsule bioshield. Elimination of the RF switch in favor of passive coupling to the relay antenna or to the diplexer transmit/receive line is under investigation. The bioshield probe allows for launch pad and pre-separation checkout of the orbiter and lander relay hardware and for relay link communications during lander separation and orientation prior to the lander de-orbit burn.

During the mission, the RRS will be operated in any one of three possible modes. The RRS will be *off* during the launch and cruise phases of the mission, operated in the *standby* mode during lander checkout, separation, entry, and during orbital phases for which relay communications are not required or feasible, and *on* during periods of required operation. In the standby mode, the RRS operates all elements except the transmitter power amplifier, thereby conserving orbiter prime power, but still providing the necessary data reception during entry. The possibility of turning the RRS *off* during the non-required orbital phases is largely dependent on achieving all necessary RRS operating characteristics with minimum warm-up. The RRS beacon transmitter would be operated during post-landed orbital passes over the lander. This is effected by CCS command to the RRS turning on the transmitter power amplifier.

f. Relay telemetry subsystem. The Relay Telemetry Subsystem (RTS) must perform the following general functions:

(1) Receive from the RRS either of two noisy PCM, split-phase waveforms (4 or 16 kbits/s).

(2) Establish synchronization between the incoming waveform and an internal VCO-driven clock.

(3) Resolve the timing ambiguity due to split-phase coding of the data.

(4) Recover the incoming data in a matched filter and restore it to binary non-return-to-zero (NRZ) form.

(5) Route the recovered PCM data to the flight data subsystem whenever the low-rate relay channel has been selected.

(6) Route the recovered PCM data and sync to the DSS at either of the two data rates.

(7) Indicate by means of a lock detector when sync is established or lost.

(8) Monitor the received SNR by sampling and processing the output of the data detector.

(9) Make up a digital status word for the flight data subsystem to be transmitted over the engineering data channel.

The major functional units of the RTS are a high-rate bit synchronizer, low-rate bit synchronizer, signal-to-noise ratio estimator, lock detectors, and power supply. A functional block diagram of the RTS (excluding the power supply) is shown in Fig. 11.

Bit synchronizer. The high-rate and low-rate bit synchronizers are functionally identical except that one operates at 16 kbits/s and the other operates at 4 kbits/s. The bit synchronizer consists of an input buffer amplifier, symbol tracking loop, bit sync ambiguity resolver, and data-matched filter.

The input to the synchronizer from the RRS is a noisy split-phase PCM waveform. The synchronizer establishes sync between the input and an internal clock, resolves

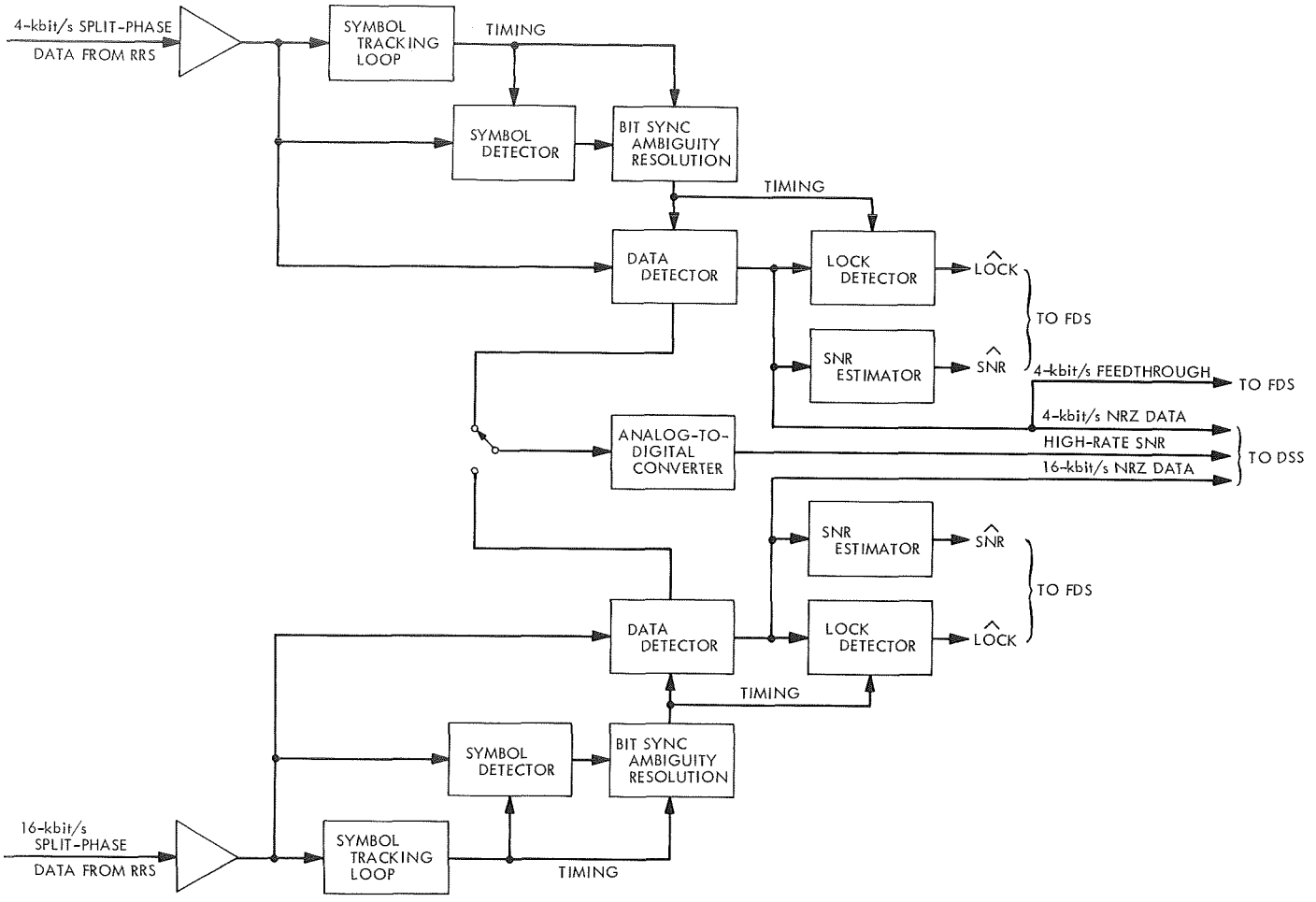

Fig. 11. Relay telemetry subsystem

the timing ambiguity, recovers the incoming data, and converts the incoming data to NRZ.

The symbol tracking loop is an analog approximation to an early–late type data transition tracking loop as shown in detail in Fig. 12. The phase detector for this mechanization consists of two multipliers, two low-pass filters (time constant of 1/3 symbol period), two absolute value circuits, and a subtractor. The subtractor output produces an error signal that controls a voltage-controlled oscillator (VCO). The VCO drives timing logic to produce various timing signals such as the early–late gate multiplying signals (A and B), dump pulse, sample pulse, and bit sync. The loop noise bandwidth ($2\ B_{L_0}$) is selected to be 2% of the bit rate.

Since the sync loop is a symbol transition loop, there will be a phase ambiguity in the bit sync timing signal. Split-phase coding has the property that mid-bit transitions exceed between-bit transitions (except for the all "1" or all "0" case at which time they are equal). The bit period phase ambiguity is readily resolved by using this property. A bit sync phase estimate is chosen to control the direction of transition counts in a 3-bit up–down counter. The bit sync phase estimate is changed only when the counter reaches full scale (or empty). The counter is inhibited from going past either full scale or empty.

Data is recovered by the use of an integrate-and-dump-type matched filter using timing from the bit sync ambiguity resolver. The input data is multiplied by bit sync (in phase with the data) and the multiplier output is integrated for a bit period. The integrator time constant is at least 4 bit periods. The output of the integrator is then fed to a decision circuit to determine whether the bit is a "1" or "0."

Lock detector. Lock detectors are implemented in both the high- and low-rate synchronizers. Their outputs will

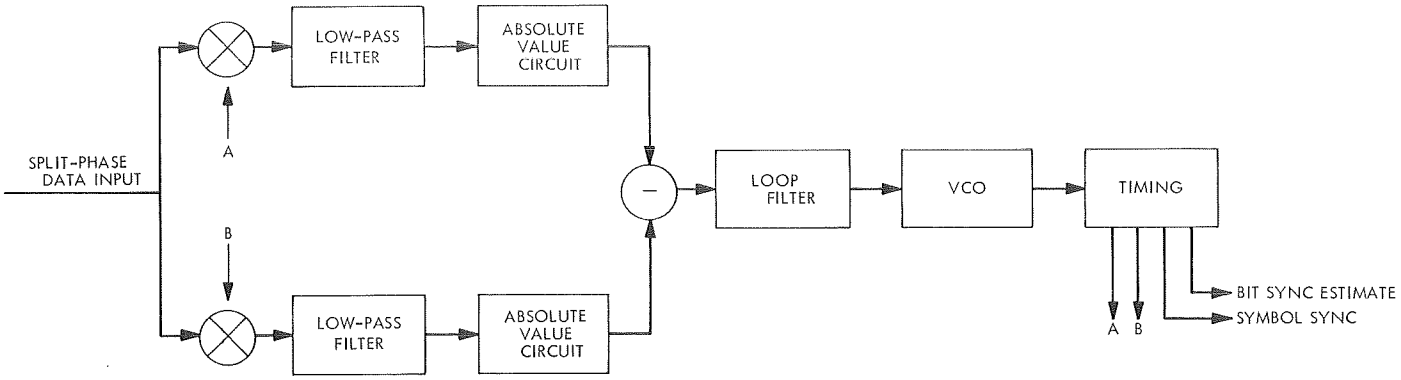

Fig. 12. Relay telemetry subsystem symbol tracking loop

be used for telemetry status information and may be used for selection of a modulation tone on the RRS transmitter. At the present time, the lock detector strategy or requirements have not been fully established. Current implementation plans are to take the absolute value of the output of the data-matched filter, average over several bit periods, and compare with a fixed threshold value. In-lock will be indicated when the fixed threshold value is exceeded.

Signal-to-noise ratio estimation. A means of estimating the relay link SNR will be mechanized as part of the RTS. Present plans call for both a high-resolution measurement to be stored on the DSS and re-transmitted in non-real time, and a lower-resolution real-time measurement telemetered via the flight data subsystem. The high-resolution measurement will consist of digitized samples of the data-matched filter output taken every N bits. Several approaches to processing the real-time SNR estimation onboard are still being considered.

Power supply. The power supply will provide the required dc operating voltages to the RTS from the 2.4-kHz, 50-V rms, square-wave power provided by the orbiter power subsystem. The power subsystem will be capable of turning *on* and *off* the power to the RTS.

Breadboard status. A functional breadboard operating at 400 bits/s is currently being tested and a functional breadboard operating at 4 kbits/s is being constructed. Tests on these breadboards are intended to provide parameter values for interfaces between the RRS and RTS and provide assurance that the implementation scheme satisfies the *Viking* orbiter mission requirements.

g. Relay antenna subsystem. The relay antenna subsystem (RAS) is required to support transmitting and receiving functions of the RRS. Study of the antenna requirements is proceeding using analyses performed for *Voyager*, Capsule System Advanced Development, and other projects. Antenna pointing and pattern development are contingent upon trajectory and mission strategy. Trajectory data for the 1975 missions are expected to be available in the last half of 1970, allowing the antenna design to be updated at that time.

Initial studies suggest use of a single radiator, circularly polarized, fixed antenna. Mechanical clearance problems may require placing the antenna on a boom in order to provide entry coverage as well as post-landed links. Trajectory data for the 1973 missions indicated that a single lobe pattern design was satisfactory for those missions. Peak gain of such an antenna would be about 4.5 dB with a beamwidth of 130 deg.

D. Guidance and Control

1. Proposed *Viking* Attitude-Control Roll-Reacquisition Logic

a. Introduction. An improvement in the functional sequences associated with the roll axis control has been designed for incorporation on the attitude-control system for the *Viking* spacecraft. The proposed sequence eliminates some unnecessary roll search modes. The attitude-control system does not immediately initiate a spacecraft roll search mode upon loss of Canopus, thus affording ground controllers time to override the automatic features of the sequence if circumstances call for it.

b. Viking spacecraft performance after loss of Canopus acquisition. The difference between the performances of the *Viking* and *Mariner* Mars 1971 systems, when Canopus acquisition is lost, is that *Viking* goes through

a timed sequence after loss of Canopus, while *Mariner* Mars 1971 does not. When *Viking* loses Canopus acquisition, while in the cruise mode, the Canopus tracker performs a flyback and sweep operation. If Canopus is not reacquired, the roll gyro turns on and the roll axis is put under inertial control. The system waits in roll inertial until a pulse is received from the spacecraft computer (an hourly pulse). When the pulse arrives, the Canopus tracker performs a flyback and sweep operation. If Canopus is not reacquired, the system waits, still in inertial, until receipt of the second pulse from the computer to flyback and sweep again. Should Canopus acquisition not result from this, the system will switch to the rate mode in roll and begin a roll search, with the roll axis under Canopus tracker control.

Mariner Mars 1971 and *Viking* have a 3-min timer which delays turning *off* the gyros after acquiring Canopus. (The gyros will not turn *off* in the presence of a condition such as straylight.) If Canopus is acquired during one of the pulse-induced flyback and sweep operations, the spacecraft will stay on inertial control in roll until the gyro turns *off*. When the gyro turns *off*, roll control switches to the Canopus tracker. Canopus acquisition has to be held for 3 min and the gyros have to turn *off* for the sequence to be terminated.

When the spacecraft is in orbit about Mars, a straylight condition will periodically occur due to reflected light from the planet. Since this straylight condition invalidates the Canopus tracker output, a signal from the spacecraft computer is sent to the attitude control, setting the roll axis to the inertial mode whenever it occurs. During straylight conditions, when Canopus acquisition is lost, the same sequence as above is activated. Whereas *Mariner* Mars 1971 would stay in inertial until acted upon by ground command, when Canopus is not reacquired, *Viking* eventually will automatically go into roll search.

This sequence can, at any time, be interrupted by ground command and the spacecraft held in inertial indefinitely or an immediate roll search initiated.

During the orbit of Mars by *Viking*, the sun will periodically be occulted. The same sequence of hourly flyback and sweeps and eventual roll search will occur after sun occultation as after straylight. The only difference is that during occultation all three axes are under inertial control. At the conclusion of occultation, the pitch and yaw gyros stay *on* until the roll gyro turns *off*.

The repeated exercise of the flyback and sweep feature in the Canopus tracker ensures that automatic reacquisition without roll search will occur if the initial loss of acquisition is due to reflected light from particles, since particle lifetime is of short duration. The minimum 1-h elapsed time between the loss of Canopus acquisition for any reason and the initiation of roll search provides time for the ground controllers to override the roll search mode should that be judged appropriate.

Figure 13 illustrates the sequence the *Viking* attitude-control system goes through when Canopus acquisition is lost. The termination of the sequence by ground command is not shown.

2. *Viking* Attitude-Control Subsystem Status Register

a. Introduction. The *Viking* spacecraft telemetry subsystem requires that there be a "status register" located in the attitude-control subsystem. This is to replace the "event register" in the *Mariner*-type spacecraft. This article describes the status register and some of the problems associated with the design.

b. Circuit description. Figure 14 is a functional block diagram of the attitude-control status register. The attitude-control logic status is input to the register in parallel. Each bit monitors the condition of a unique logic state in the attitude-control subsystem. At a telemetry bit rate of 33⅓ bits/s, the telemetry system reads the attitude-control status word approximately once every 7 min. At the time that telemetry desires to read the attitude-control status word, the alert pulse line is enabled. This inhibits the inputs from the attitude-control logic and converts the register to shift-right operation. Clock pulses then transfer the status word to telemetry bit by bit.

The preliminary bit assignments for the attitude-control status register are as follows:

Bit 1	Gyros *on* (pitch, yaw, and roll)
Bit 2	Roll gyro *on*
Bit 3	Sun acquired yes/no
Bit 4	Stray light yes/no
Bit 5	Sun occultation yes/no
Bit 6	Star acquired yes/no
Bit 7	Autopilot *on*/gain change yes/no

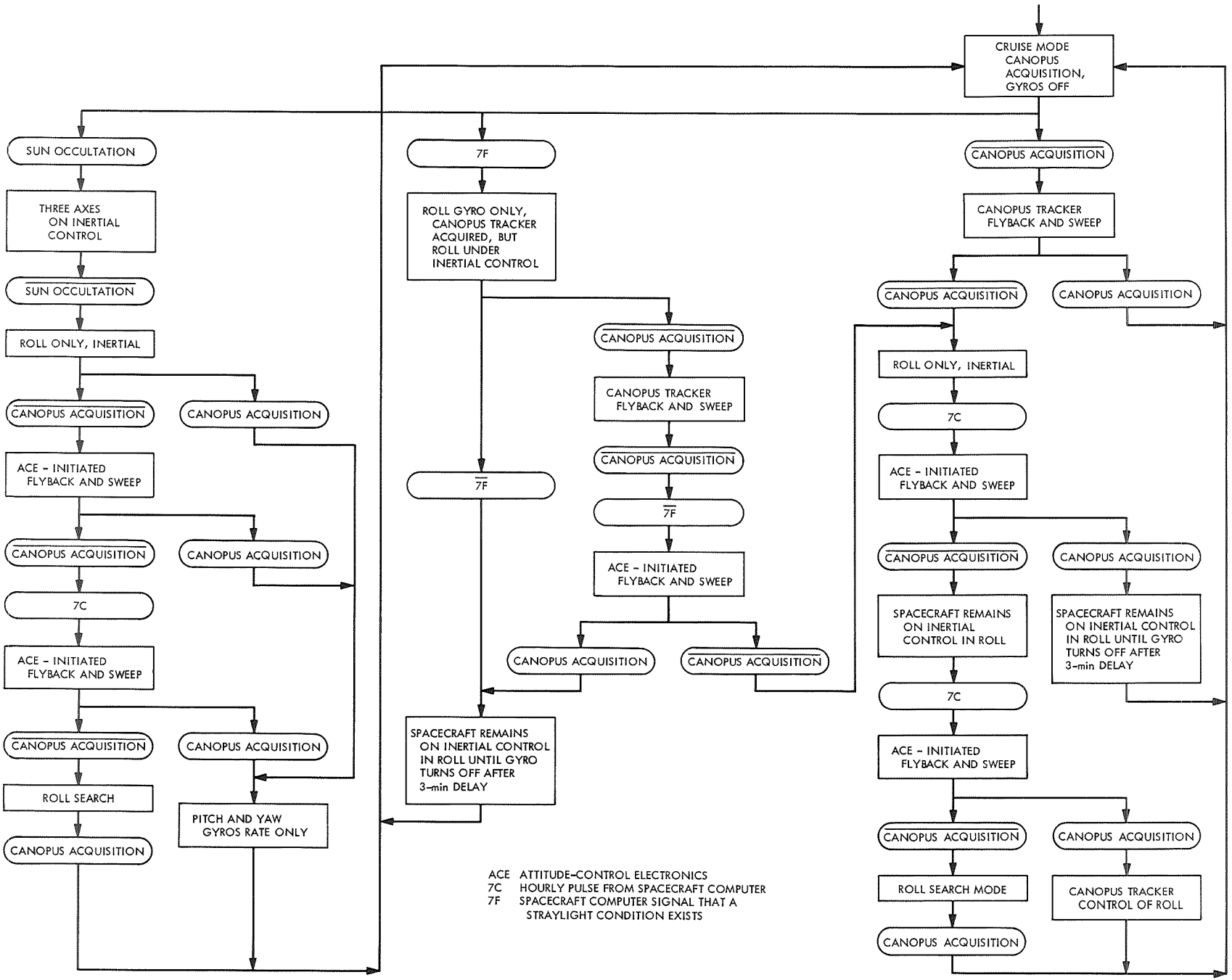

Fig. 13. *Viking* attitude-control sequence for Canopus reacquisition

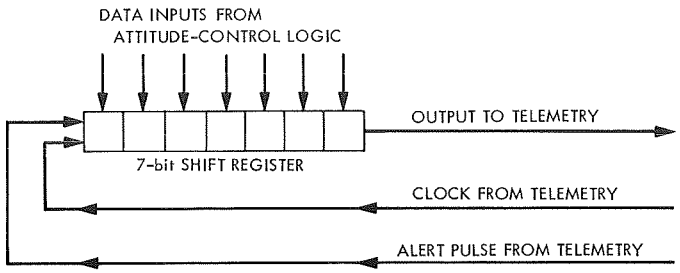

Fig. 14. *Viking* status register functional block diagram

These assignments are not final. It may be preferable to monitor the state of some of the logic switches internal to the attitude-control electronics since some of the functions listed above can be readily determined by telemetry in the high-rate channels.

c. Design considerations. Two designs of the status register are presented in Figs. 15 and 16. Figure 15 shows the design of a status register using Signetics 100-400 diode-transistor-logic (DTL) microcircuits. This family of microcircuits uses a B+ voltage of 4.5 V. Figure 16 shows a status register design using Texas Instruments 54L low-power microcircuits. This family of microcircuits uses a B+ voltage of 5 V. A comparison of alternate design approaches is shown in Table 4.

The table clearly shows the advantages of using low-power microcircuits. It should be noted that Texas Instruments 54L series is used in telemetry; if it is decided to use the Signetics 100 series in attitude control, additional circuits may be required to interface between the telemetry low-power devices and the attitude-control high-power devices.

d. Conclusions. Both status register designs will meet the design requirements of the *Viking* spacecraft. In terms of number of flatpacks and power consumption, the low-power design is clearly superior. However, other considerations, such as the available B+ voltage (4.5 V vs 5 V), may dictate which design is ultimately used.

3. *Viking* Scan Subsystem Mechanization

A modification to the *Mariner* Mars 1971 scan control that results in some hardware economies and in improved operational flexibility is proposed for the *Viking*.

The *Viking* mission imposes new requirements on the scan subsystem that cannot be met without changes to the existing hardware. A platform slew rate of 1 deg/s with a step resolution of 0.25 deg/step is required. The

Table 4. Microcircuit evaluation

Device	B+ voltage, V	Number of flatpacks	Power, W
Signetics 100-400 series	4.5	13	~0.5
Texas Instruments 54 series	5.0	3	~0.5
Texas Instruments 54L series	5.0	3	~0.06

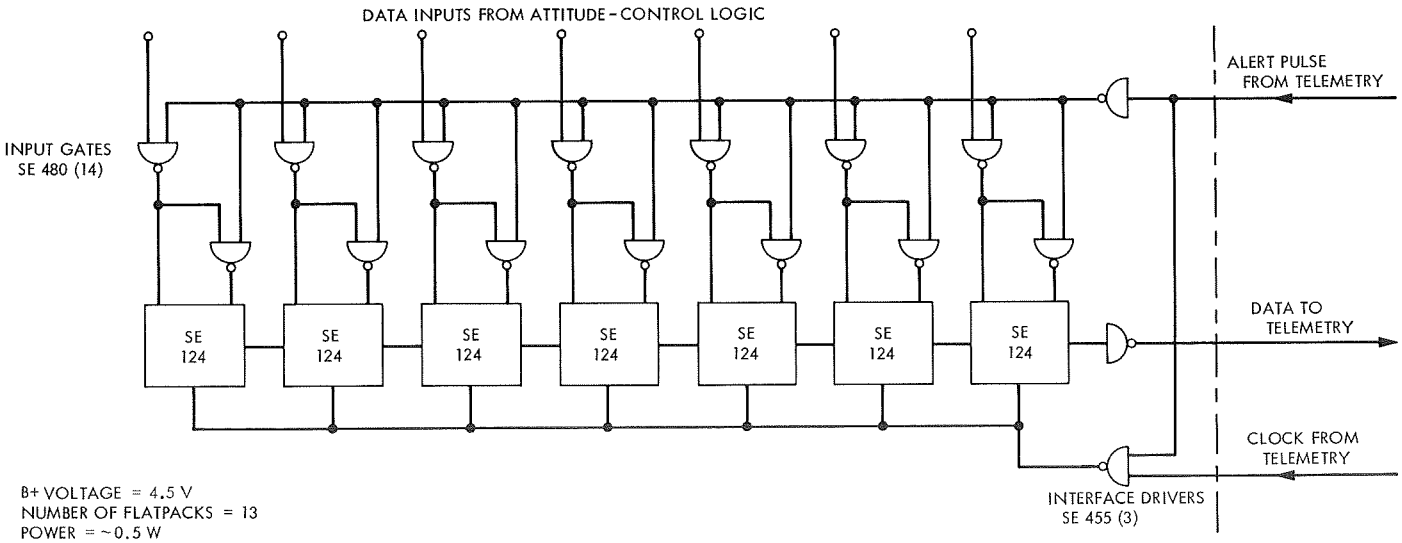

Fig. 15. *Viking* attitude-control status register preliminary design using DTL microcircuits

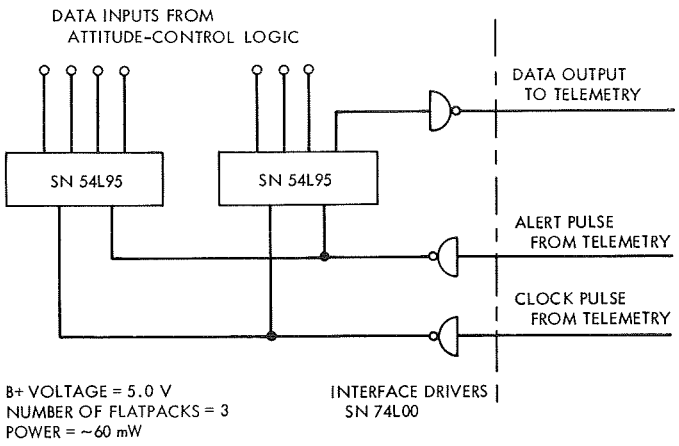

Fig. 16. *Viking* attitude-control status register preliminary design using low-power microcircuits

Mariner Mars 1971 hardware has a 1-deg/s slew rate with a step resolution of 1 deg/step.

Figure 17 is a block diagram of the scan subsystem. The shift register and the two digital-to-analog converters serve as memory devices in lieu of the previously used stepper motor-driven potentiometers. A 20-bit word corresponding to the angular clock and cone positions is shifted serially into the register. The two 10-bit digital-to-analog converters generate two analog voltages corresponding to cone and clock angular positions. Each converter has a dynamic range of 1024 and assuming that a 0.25-deg resolution is implemented, this results in a total range of the platform of 256 deg per axis. A correspondingly higher total range can be obtained at the cost of angular resolution. A two-line interface with the computer command subsystem (CCS) is assumed consisting of an information line and a clock line. This interface is very similar to the *Mariner* Mars 1971 ACE–CCS (attitude-control electronics–computer command subsystem) autopilot pre-aim interface. A coded command backup mode to perform identically to the CCS interface is shown in Fig. 17.

The platform actuator is slaved to the digital-to-analog converter output via a positional servo. The slewing capability of this system is independent of its angular resolution characteristics and depends only upon the actuator and load properties of the subsystem. Assuming these to be similar to the *Mariner* Mars 1971, a nominal slew speed of 1.3 deg/s can be achieved.

This proposed *Viking* mechanization eliminates the need for a pressurized electromechanical module containing stepper motor-driven potentiometers and results in the saving of approximately 20 electronic piece parts over the existing scan subsystem design. The proposed usage of the shift register digital-to-analog combination in contrast to the existing design results in a volatile memory. Because of the difference in missions between the *Viking* and the *Mariner* Mars 1971, the non-volatile memory is no longer a requirement.

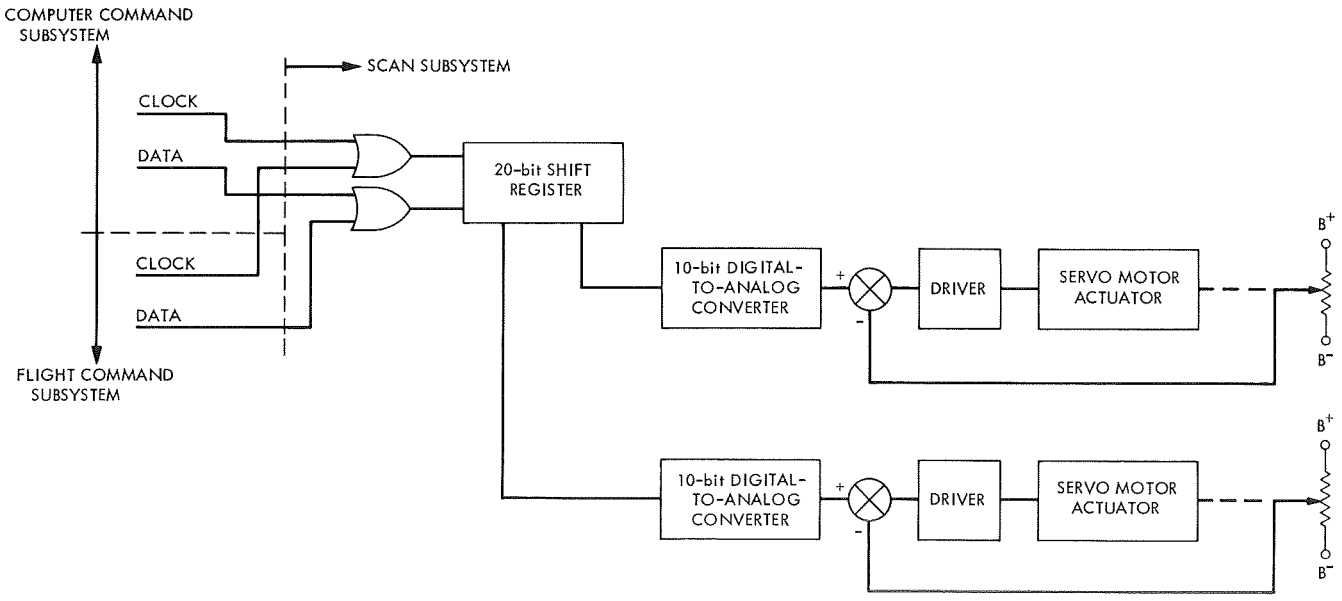

Fig. 17. *Viking* scan subsystem